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FLIGHT TEST OF A PITCH CONTROL FOR A SPINNING VEHICLE

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SUMMARY

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A flight test was made of a simple single-axis attitude control system for the fourth stage of a spin-stabilized vehicle. The system utilized two body-fixed horizon detectors for attitude sensors, a single rate gyro for stability augmentation, and a single jet for control, and it weighed less than 18 pounds including batteries and control-jet gas. Flight results showed the attitude error to be less than $\pm 1/2^{\circ}$ and the coning angle to be less than $\pm 2^{\circ}$ throughout the test period. Very close correlation was obtained between actual flight-test results and preflight computer studies.

INTRODUCTION

Spin stabilization is a widely used method for maintaining the attitude of a vehicle in space. This presentation describes the results of the flight test of a control system that makes use of the characteristics of the spinning vehicle to provide an extremely simple pitch-attitude control system.

Reference 1 describes three configurations of attitude control systems for spinning vehicles - horizontal or near horizontal, vertical, and angle of attack. The system that was flight tested is a variation of the horizontal or near horizontal scheme, adapted to fit the conditions imposed by a specific application. These conditions required the system to damp coning rates to less than $\pm 6^{\circ}$ per second, and reach steady-state pitch attitude to within $\pm 1/2^{\circ}$ within 10 seconds of control initiation, operation of the control system to begin at ignition of the final-stage motor. Coning rate is defined as the vector sum of the rates about the vehicle pitch and yaw axes. Absolute pitch-attitude reference was derived from body-fixed horizon detectors, and damping commands were derived from a rate gyroscope. The attitude corrections were made by a single body-fixed reaction control jet operating on stored nitrogen. The system used only 2 moving parts and 13 transistors, and its total weight, including batteries and nitrogen, was less than 18 pounds.

MAJOR PRINCIPLES OF OPERATION

The control system, shown in figure 1, used two body-fixed horizon-detecting telescopes for attitude reference, a single rate gyro for damping, and a single jet to produce the control torque. In operation, telescopes F and S ("forward" and "side") provide signals to the electronics each time the earth is seen. These signals are amplified and operate an electronic switch which turns on the control jet whenever an earth signal is present. The control jet and telescopes are positioned on the body in such a way that when telescope F sees the earth the vehicle is torqued in the nose-up direction, and when telescope S sees the earth the vehicle is torqued in the nose-down direction. In addition, a rate-gyro signal is fed into the same electronics and operates the same control jet to provide artificial damping to the system. A detailed description of this system will be found in reference 1.

The system used differs from that described in reference 1 because of the presence of the free jet, or plume effect, from the burning rocket motor. The presence of this free jet precludes aiming telescope S at such an angle as to see the rear horizon of the earth. Forward telescope F, then, is used as the attitude-reference telescope, and when F fails to see the earth for one revolution of the vehicle, side telescope S is used to provide a coarse correction in a direction that will bring the scan of telescope F back on the earth.

PREFLIGHT SIMULATOR STUDIES

Trajectory studies of the uncontrolled portion of the vehicle flight provided the initial conditions expected for control initiation at fourth-stage ignition. These conditions were an altitude of 235,000 feet, pitch angle of 55° with respect to local horizontal, a representative command pitch attitude of 65°, and a spin rate of 3 rps ± 0.5. In addition to these conditions, a tip-off disturbance resulting from final-stage ignition and separation was anticipated, and previous experience with similar vehicles indicated that this disturbance could result in initial coning rates as high as 36° per second.

These initial conditions and the moments of inertia of the last stage of the vehicle were included in an analog-computer program similar to that described in reference 1, and the computer signals were used to provide signals to the actual vehicle electronics and control-jet solenoid valve. The solenoid valve mechanically operated a switch to provide the electrical analog of thrust to feed back into the computer. Signals for the horizon-detector telescopes were provided by utilizing the appropriate computer signals to light small lamps mounted in the telescopes, and the rate-gyro signal was provided by modifying the appropriate computer signal by simulated rate-gyro dynamics. Using the actual hardware in this fashion insured correct values of time lags, dead spots, hysteresis, and other such nonlinearities.

A rate-gyro signal threshold of 4° per second was found to allow a rapid enough system operation to achieve steady-state conditions within 10 seconds and still keep the coning rate within the required 6° per second. The addition of a 1-second damper mode at initiation of control, to damp out the initial disturbance before providing attitude control, decreased markedly the control-jet fuel usage for cases of large initial disturbances. Without the initial damper-only mode, large tip-off rates may cause a condition in which the rate and attitude signals

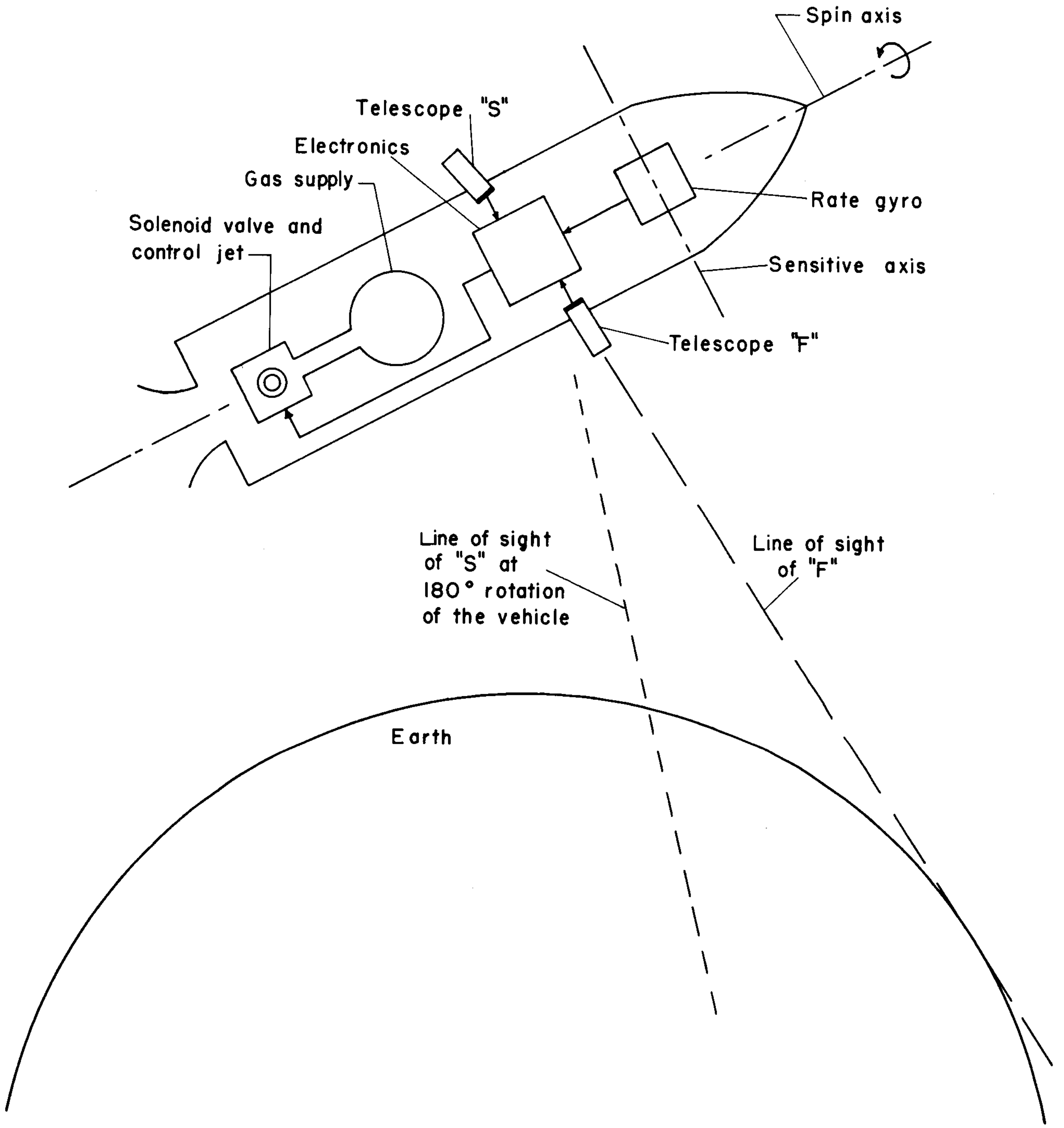

Figure 1.- Control-system configuration and block diagram.

tend to fight each other and cause the jet to blow almost continuously, which contributes little to attitude control or damping. An additional saving of control-jet fuel was accomplished by offsetting the position of the jet about the spin axis with respect to the telescopes and rate gyro. This angle of offset is determined by the design spin rate and the solenoid-valve time lag, and provides a first-order compensation for the valve time lag.

With these modifications to the basic system, steady-state conditions were reached in 10 seconds or less for all combinations of initial conditions in attitude, coning rate, and spin rate studied. According to the studies, requirements for the control jet were a nominal 40 pounds' thrust at a pressure of 3,000 psi and a location 34 inches from the center of gravity of the vehicle. The nitrogen supply for this jet was determined to be 150 cubic inches at 3,000 psi unregulated, so that the thrust of the jet varies with the amount of gas which has been used.

The computer showed the steady-state coning angle to be $\pm 1\frac{1}{2}^{\circ}$ to $\pm 2^{\circ}$ at the threshold rate of 4° per second and the axis of this cone to be within $\pm 1/4^{\circ}$ of the command point at the nominal spin rate of 3 rps.

FLIGHT SYSTEM

Vehicle

The vehicle chosen (shown in fig. 2) was a four-stage configuration commonly called a Javelin, consisting of an Honest John first stage, Nike second and third stages, and an Altair fourth stage. The first two stages were aerodynamically

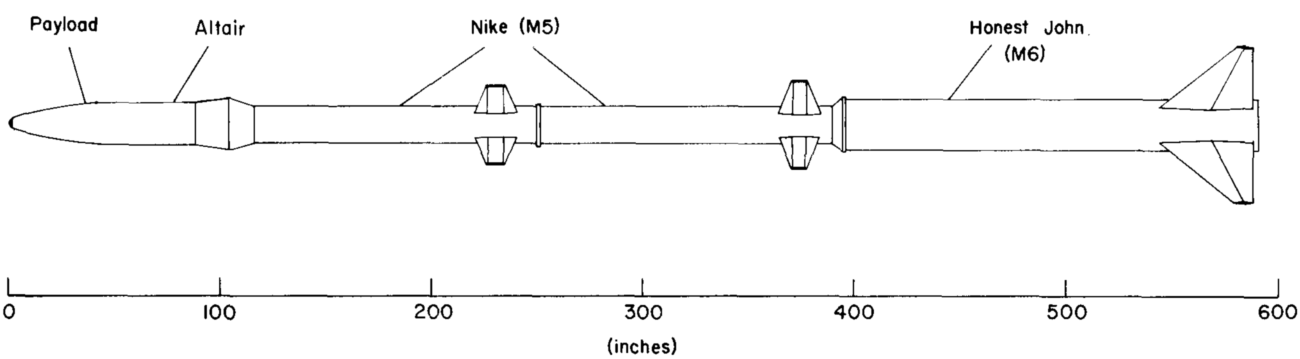

Figure 2.- Four-stage vehicle configuration used for flight test.

stabilized by large fins on each stage, and used a drag separation system. At second-stage separation, canted fins on the third stage caused an aerodynamic spin-up that provided stability during atmospheric exit. The third and fourth stages were separated by a blast-diaphragm separation system activated by ignition of the fourth-stage motor.

The payload section of the fourth stage housed the control-system electronics and power supplies and the telemetry and associated power supplies. The control jet, valve, and reaction-gas tank were housed in the flare at the aft end of the fourth stage. The moments of inertia of the fourth stage with the loaded rocket motor were 60 slug-ft^2 in pitch and yaw and 6.5 slug-ft^2 in roll.

Trajectory

Six-degree-of-freedom trajectory studies predicted that the aerodynamic moments on the spinning third- and fourth-stage combination after burnout of the third-stage rocket motor, combined with the rapidly changing flight-path angle, would cause the vehicle to precess in the yaw and pitch planes. Range safety considerations limited the amount of precession that could be tolerated, which in turn set the conditions for ignition of the fourth stage.

The conditions for ignition of the fourth stage then were an altitude of 235,000 feet and a pitch-attitude angle of 55° with respect to the local horizontal. The spin rate at ignition was approximately 3.5 rps. The flight test was specified to be terminated by an automatic destruct 12 seconds after ignition of the fourth stage in order to comply with range safety requirements.

Control Components

Electronics.- The basic purpose of the control electronics is to switch on the control-jet solenoids whenever a horizon signal is detected by either telescope or when a rate signal exceeding a preset value is received from the rate gyro. For this particular application an "inhibiter" circuit prevents signals from telescope S from reaching the switching circuitry for a period of about 0.3 second each time a signal is received from telescope F. This device insures that as long as a horizon signal is received from telescope F at each revolution of the vehicle about its spin axis, indicating a nose-down error, signals from telescope S will be ignored, but if telescope F fails to see the horizon at about the proper time in the spin revolution, indicating a nose-up error, signals from telescope S will be allowed to activate the jets, bringing the nose down until telescope F sees the horizon.

Figure 3 shows the complete control-system electronics except the telemeter coupling networks, ground switching and monitoring circuitry, and power supply. The diode logic network allows signals from either of the two telescopes or negative signals from the rate gyro to enter the control amplifier, which employs a field-effect transistor for high input impedance. The amplified signals operate an electronic switch and power amplifier, which actuates the jet control valves directly.

The inhibiter consists of an amplifier similar to the control amplifier and a special monostable multivibrator which may be retriggered at any time. The multivibrator is triggered by amplified signals from telescope F each time it sees the horizon. When triggered, it closes a diode gate in the input circuit from telescope S for a period of about 0.3 second or one spin revolution.

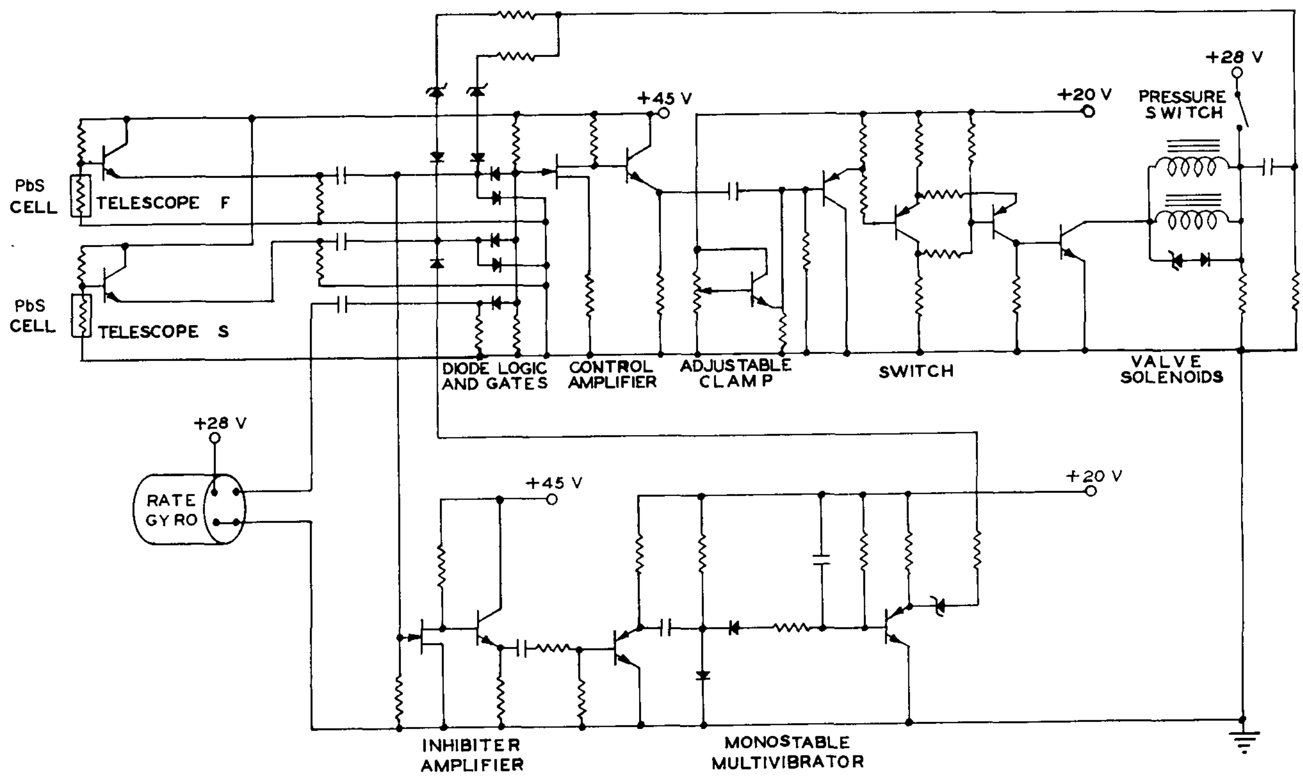

Figure 3.- Control-system electronics and wiring diagram.

Power is supplied to the entire system except the solenoid valves before launching. Power is supplied to the solenoid valves by a pressure switch on the fourth-stage rocket casing at the time of firing to initiate control action. A capacitive network energized by this switch closes diode gates in both telescope circuits for about 1 second to allow rate signals from the gyro to subdue any large initial tip-off rates before attitude correction is started. The rate threshold, above which rate signals actuated the control jet, was set at 4° per second. Figure 4 shows the electronics assembly (labeled "C").

Horizon telescopes.- The horizon telescopes (labeled "A" in fig. 4) employed simple quartz optics of 3/8-inch aperture and $2\frac{1}{2}$- inch focal length to focus radiation on a lead sulfide photodetector whose spectral response was from ultraviolet through 2.5 microns. The total field of view was approximately 1°. Individual emitter followers were installed in each telescope housing to reduce noise pickup in the connecting cables. Telescope F is aimed 16.4° forward of a plane normal to the body spin axis, and telescope S is aimed 4° aft of this plane. These angles were chosen for this particular case and would be varied to suit other vehicle pitch-angle requirements.

Rate gyro.- Since the vehicle rates to be measured were two orders of magnitude less than the spin rate, some care was required in the selection and installation of the rate gyro. The gyro selected (labeled "B" in fig. 4) had rugged ball-bearing gimbal supports to withstand the excessive precession loads imposed by vehicle spin. It was mounted in the vehicle with the output axis parallel to the vehicle spin axis to minimize the effect of vehicle spin velocity on the gyro input axis. Interaction due to mounting inaccuracies was minimized by adjusting the mounting bracket for zero gyro output when the vehicle was being spun up on the dynamic-balancing machine.

The gyro housing contained its own inverter and demodulator so that it would operate directly from the 28-volt control power and produce a direct-current output signal compatible with the control electronics.

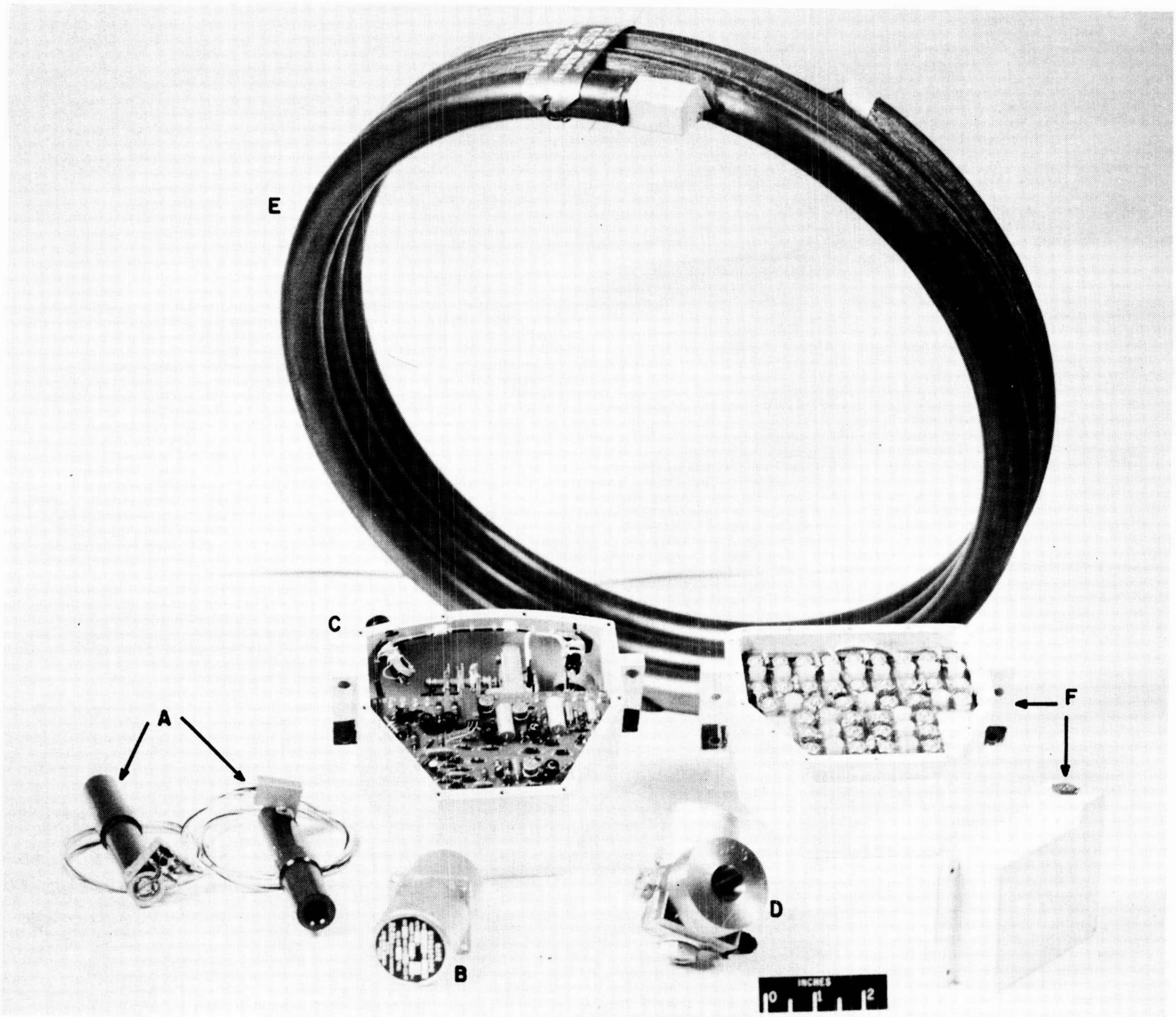

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Figure 4.- Complete control system. A - Telescopes; B - Rate gyro; C - Electronics; D - Valve and jet; E - Control-jet gas tank; F - Batteries.

Control jet and valve.- The control jet was required to produce a thrust of 40 pounds at the initial supply pressure of 3,000 psi. Rapid operation of the solenoid valve controlling the jet was necessary to avoid excessive phase lags. Since the only solenoid valves which could be found with acceptably low operating times did not have a large enough opening to supply a 40-pound jet, two identical 20-pound jets and solenoids were operated in parallel. The valves were adjusted to have minimum opening and closing times of approximately 0.018 second. Figure 4 shows one of the solenoid valve and jet assemblies (labeled "D").

Pressure-supply tank.- The tank supplying pressurized nitrogen to the reaction jets (labeled "E" in fig. 4) consisted of a helical coil of steel tubing. The storage volume was 150 cubic inches and the initial pressure was 3,200 psi.

Power supply.- The 28-volt power supply for the control system was provided by small silver-zinc storage cells. These supplied power to the rate gyro and the solenoid valves. Simple zener diode regulators supplied the 20-volt requirements of the electronic circuitry. Dry cells were used for the 45-volt supply. Power requirements were so modest that the internal batteries were sufficient for all preflight ground checks. Battery supplies are shown in figure 4 (labeled "F").

FLIGHT-TEST RESULTS

Figure 5 is a time history of command attitude and vehicle pitch attitude, starting at control initiation. The command-attitude change shown is due to the increase in the depression angle to the horizon caused by the changing altitude of the vehicle. The initial pitch attitude of the vehicle at separation was about 6° higher than the trajectory studies had indicated, giving an initial pitch command of only 4°. Steady-state conditions were reached at about 5 seconds, with a coning angle of $\pm 1\frac{1}{2}^\circ$ to $\pm 2^\circ$ as predicted by the preflight computer studies and an average error of less than $\pm 1/2^\circ$ in pitch attitude.

The pitch attitude was determined from the signals from a third telescope which was aimed 7.8° aft of the forward control telescope. This telescope remain on the earth during the coning period when the forward control telescope is off the earth, and allows pitch-attitude measurements to be made once in each revolution of the vehicle about the spin axis.

An unexpectedly small tip-off disturbance resulted in less control-fuel usag than was anticipated. This, coupled with the failure of the destruct system to destroy the model, allowed the control period to extend well beyond the design time of 12 seconds.

In this flight test no control was provided in the yaw plane. Deviation in yaw was due to aerodynamic moments applied during the coast phase, to tip-off moment, and to cross-coupling between the pitch and yaw axes contributed by the damping portion of the control system. The deviation was equivalent to 15° of vehicle yaw, applied at ignition of the fourth stage.

DISCUSSION OF FLIGHT-TEST RESULTS

In figure 5 the design portion of the operation of the control system - the first 12 seconds - shows excellent correlation with the preflight computer studies of the system. The instantaneous attitude of the vehicle in the pitch plane shows that the coning angle of the vehicle was essentially the same as that predicted by the computer studies, and the average attitude - the center line of the cone - was within $\pm 1/2^{\circ}$ of the set point, as compared with $\pm 1/4^{\circ}$ for the computer study.

Figure 6 is a reproduction of the telemeter records for the first 7 seconds of actual flight after control initiation. During the initial portion of the records the system is in the damper mode and the signals from telescopes F and S are inhibited by the control delay circuit. This is evidenced by the "soft" appearance of the signals from F and S. The initial coning rate is below the gyro-channel threshold. The partial signals on the solenoid-valve channel are caused by noise on the gyro signals which is of too high a frequency to appear on the telemeter records and is not of a high enough level to operate the valve. The signal pulses on the telescope F channel are seen to become sharp beginning with the third pulse, as the control delay circuit ceases to function. However, the signals from F are not at their full amplitude during the third and fourth pulses, as the control delay circuit still exhibits a slight influence on the signal. For these two pulses, the signal to the solenoid valve is seen to be erratic, indicating that the signal level from F is occasionally falling below that required for operation of the valve. From an analysis of these records, it was concluded that the maximum signal from F is only about 20 percent above

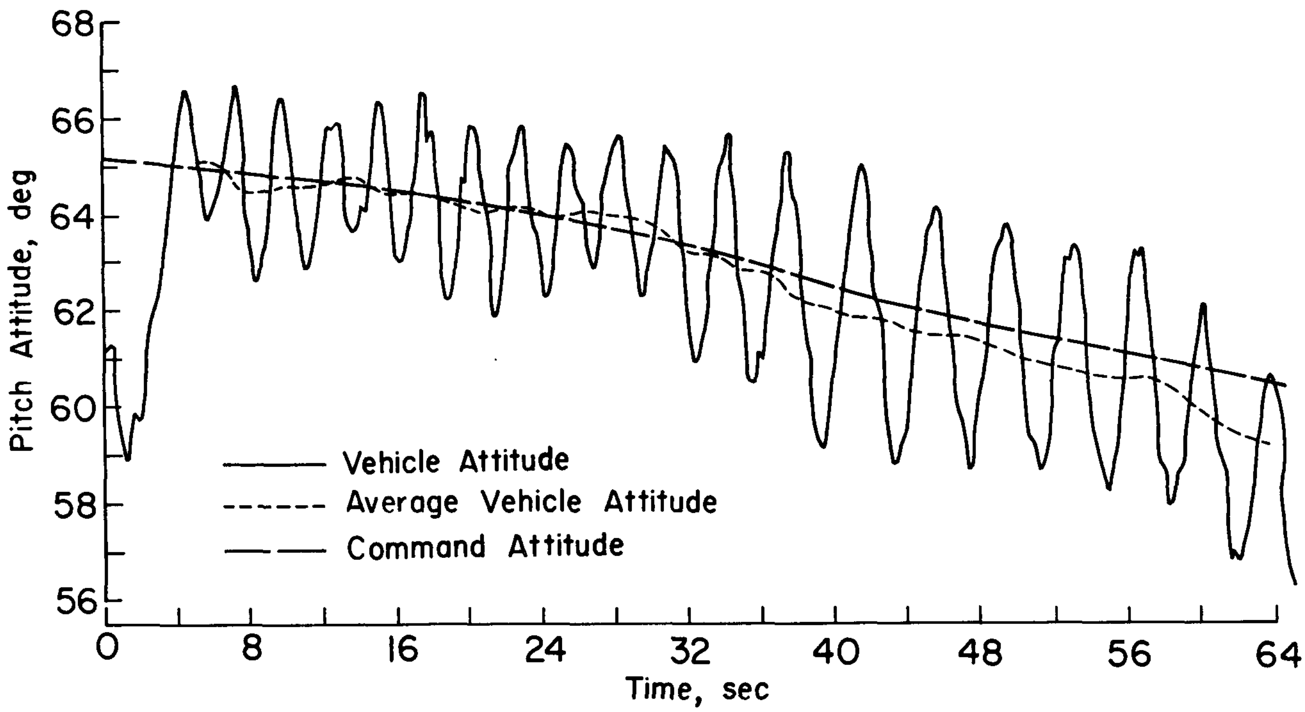

Figure 5.- Pitch attitude as a function of time after control initiation during flight test.

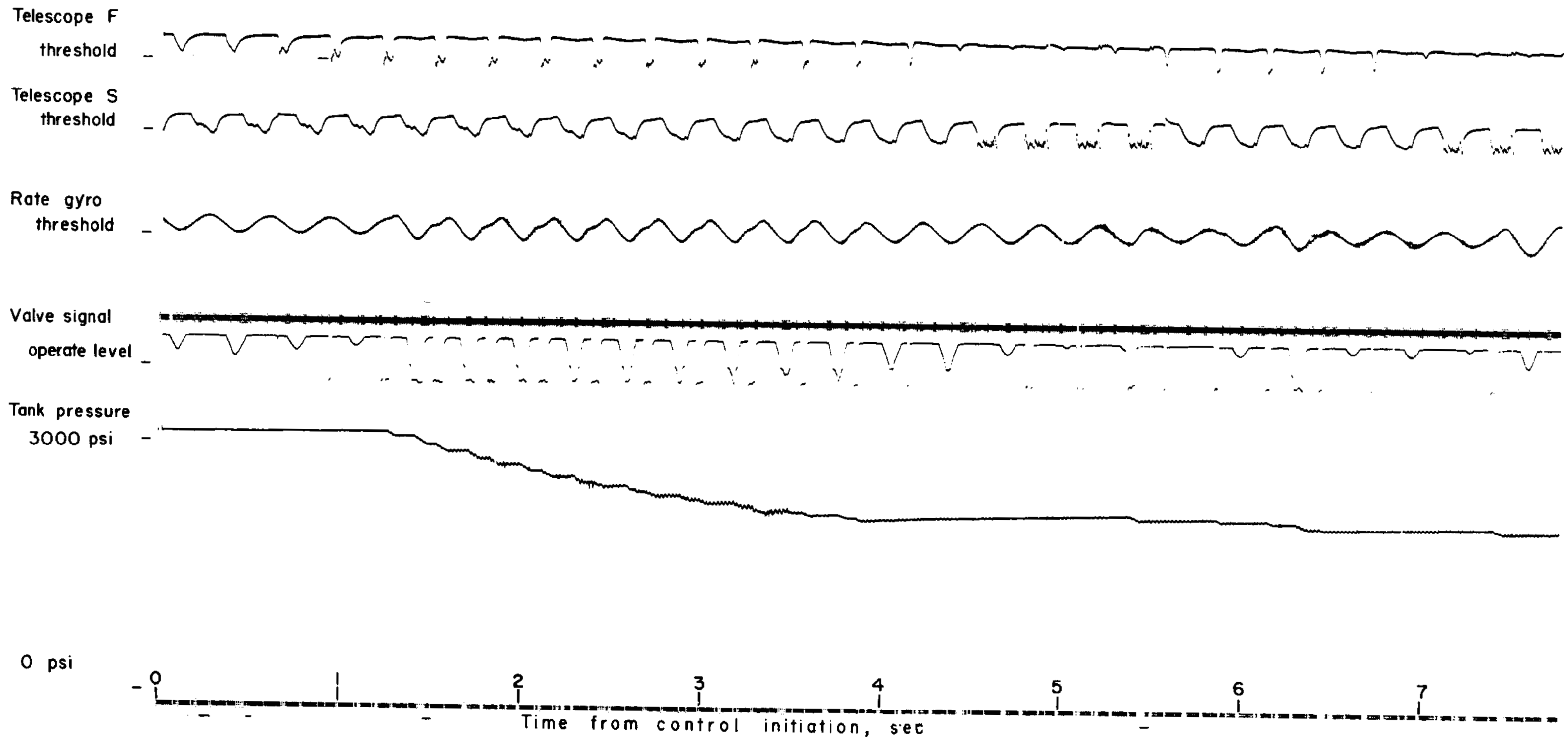

Figure 6.- Telemeter records of initial 7 seconds of control operation.

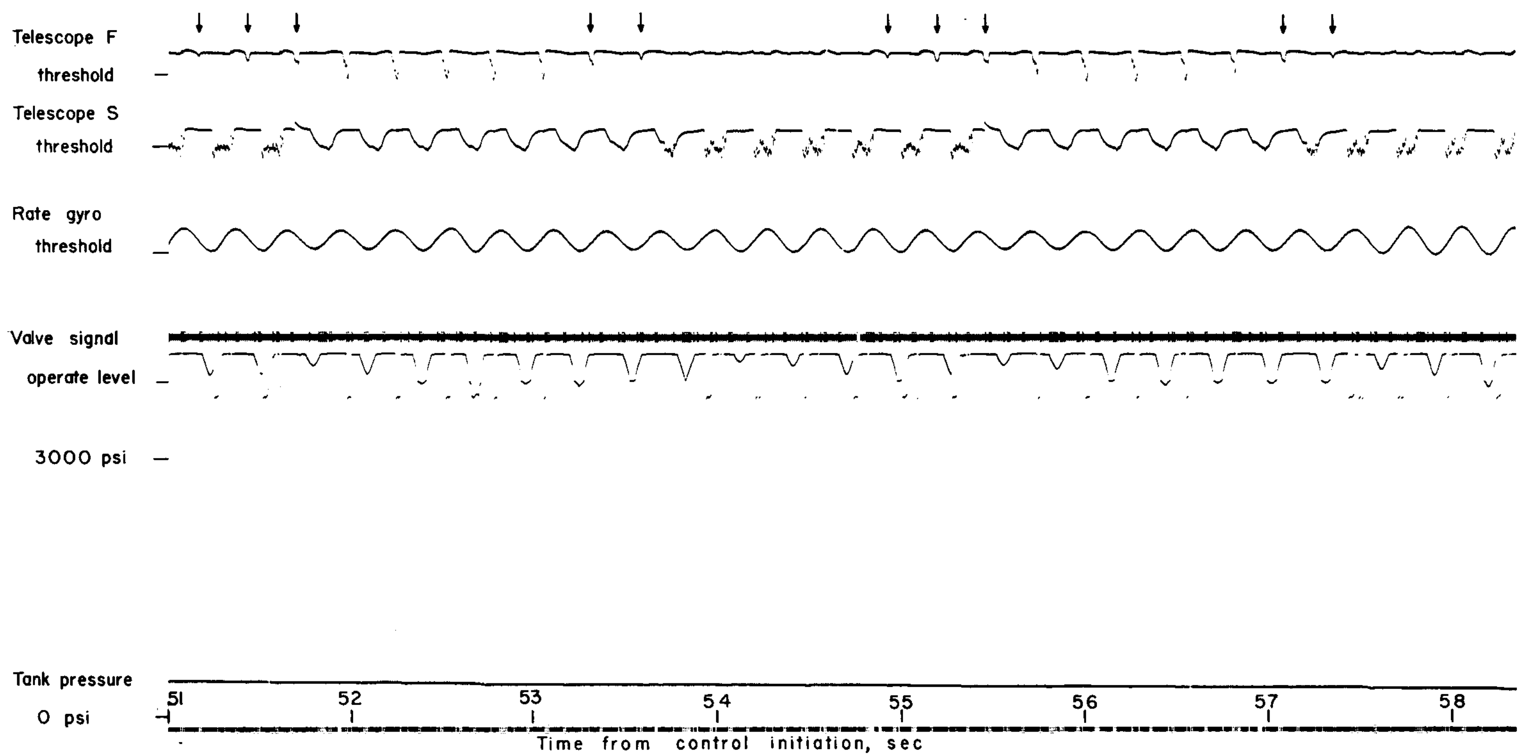

Figure 7.- Telemeter records at 51 to 58 seconds after control initiation. Arrows on telescope F channel denote horizon signals of insufficient amplitude to provide system operation, causing a change in effective horizon angle.

the minimum signal that will operate the solenoid valve. This is borne out by the erratic operation of the valve by telescope S, which was found during preflight calibration to have approximately 80 percent of the gain of telescope F. As a result the signal level continued to be marginal in this channel.

At about 35 seconds (fig. 5) the vehicle attitude begins to fall below the command attitude. Examination of the telemeter records shows this to be due to the very marginal gain in the horizon-detecting telescope channels and the chang in contrast between space and earth. At this time, the vehicle has attained suf ficient altitude and has precessed toward the north far enough for the line of sight of telescope F to be almost directly away from the sun when looking at the forward horizon. This condition produces an effect of limb darkening, which reduces the scattered sunlight seen by the telescopes. When the signal at the forward horizon is reduced to 80 percent of the value it had near the beginning of the data period, the signal is not sufficient to operate the system and the vehicle is torqued down until the signal from F is large enough to actuate the system. This action creates a false horizon somewhat lower than the real horizo and accounts for the nose-down error between the control attitude and command attitude starting at 35 seconds on figure 5.

Figure 7 is a reproduction of the flight telemeter record between 51 and 58 seconds. The effect of the limb darkening can be seen in the reduced signal from telescope F indicated by the arrows and the fact that these signals are not large enough to inhibit the signal from S or to actuate the solenoid valve.

CONCLUDING REMARKS

Results of this flight test demonstrate the feasibility of controlling a spinning vehicle with a simple pitch-attitude control system that uses body-fixe horizon detectors for an attitude reference, a single rate gyro for stability, and a single jet for control. The final steady-state attitude error in the control plane and the size of the precession cone were essentially as predicted by preflight analog-computer studies of the system.

The system which was flight tested was a single-axis control system. However, analog-computer studies and preliminary hardware design show that the system can be extended to a two-axis control by use of a suitable yaw or azimuth reference such as a sun detector. This would also be a body-fixed detector and would use the same amplifier and control jet as the horizon detectors and rate gyro. Hence the extension of this system to a two-axis control suitable for suc missions as orbit injection would not increase the number of moving parts and would add very little to the electronic complexity of the system.

Langley Research Center,
National Aeronautics and Space Administration,
Langley Station, Hampton, Va., April 18, 1963.

REFERENCE

1. Garner, H. D., and Reid, H. J. E., Jr.: Simulator Studies of Simple Attitude Control for Spin-Stabilized Vehicles. NASA TN D-1395, 1962.